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RESEARCH MEMORANDUM

SOME DESIGN IMPLICATIONS OF THE EFFECTS OF AERODYNAMIC HEATING

By Richard R. Heldenfels

Langley Aeronautical Laboratory
Langley Field, Va.

NASA LaRC 6-16-83
1/98

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON
July 28, 1955

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SUMMARY

The structural design problems created by aerodynamic heating appear to be numerous, complex, and very severe. Two of these problems, creep and thermal buckling, are examined to indicate their effect on the design of structures for high-speed aircraft; then consideration is given to the use of insulation as a means of alleviating the effects of aerodynamic heating. The results show that creep may not be a significant factor, but thermal buckling may have a substantial effect on the structural design. The use of insulation has merit under certain conditions but poses many new design problems. These results are based on limited data and may change as more information becomes available.

INTRODUCTION

Many papers in the recent literature (for example, refs. 1 to 8) have discussed the various structural effects of aerodynamic heating. These papers have made it clear that the resulting structural design problems are numerous, complex, and so severe that the performance capabilities of high-speed aircraft may be greatly restricted. A more optimistic view of the situation, however, indicates that the severity of most of these problems is often overemphasized. The purpose of this paper is to take a brief look at the effect of two problems, creep and thermal buckling, on the design of aircraft structures and to give some consideration to the use of insulation as a means of alleviating the effects of aerodynamic heating.

The symbols used are defined in appendix A.

CREEP

The creep of aircraft structures at elevated temperature is often assumed to be a very important problem. Its importance, however, has not been established, and before accepting it as a major problem, its significance in aircraft design should be examined.

Creep is a primary design criterion in much high-temperature machinery, such as heat exchangers and gas turbines, but in this type of equipment the material must withstand the combined design temperatures and loads for long periods of time. The usual aircraft structure on the other hand is subjected to a variety of loads that occur under varying temperature conditions.

The relationship between load and time typical of present-day airplanes is shown in figure 1 where the load ratio n/n_L is plotted against the percentage of total flight time spent above that load ratio. The load ratio is the actual load n divided by the design limit load n_L. The solid line is for a fighter-type airplane and the dashed line for a bomber type. This figure was prepared from existing gust-loads data (ref. 9) and the generalized maneuver-load curves presented in reference 10. The time spent at the low load ratios for each type is primarily the result of gusts and that at the higher loads is primarily due to maneuvers. Note that percent time is plotted on a logarithmic scale and that most of the total time is spent at low load ratios, about 90 percent of the time for the fighter and 99.9 percent for the bomber.

The foregoing results were from airplanes in flight at speeds below sonic. It is a matter of conjecture what similar relations will be for supersonic airplanes, but it seems reasonable to assume that the load-time relations will not be substantially different from those shown in figure 1. It is significant, then, that high-speed airplanes will spend a very small percentage of their lifetime at loads near the design limit load where creep is most likely to be important. In addition, these high loads will not necessarily occur in conjunction with high temperatures.

The uncertainties that exist in the expected loads and the limited data available on cyclic and intermittent creep make a detailed analysis of the effect of creep on structural weight impractical at present, but some approximate indications can be obtained as follows:

Let all the loads be represented by only two load levels, a low level representative of most of the flight time (for example, about 0.2 load ratio for 90 percent of the life of the fighter) and then assume, very conservatively, that the remainder of the life is spent at the design limit load. The lower load level then makes an insignificant contribution to the creep of the structure. The lower load level is more important for the bomber than for the fighter because it is a large percentage of the design load and because the bomber would have a longer expected lifetime, but still the lower load ratio makes only a small contribution. For either case, the effect of creep on structural weight can be estimated very conservatively by fixing a lifetime of 100 hours at temperature and limit load combined. Results of such an analysis are shown in figure 2 where the weight required to support a given constant tensile load is plotted against temperature for three structural materials and for two design criteria. The solid curves indicate the weight required when the criterion is the ultimate tensile strength; the dashed lines give the weight required when creep rupture at limit load is used. The three materials selected for this analysis, 2024-T3 (24S-T3) aluminum alloy, RC-130A titanium alloy, and Inconel X, cover the temperature range in which metals are usable. The stainless steels generally fall between RC-130A and Inconel X, but nearer RC-130A.

Consider first the aluminum alloy. The weight required to support the load on the basis of ultimate tensile strength $1\frac{1}{2}$ times the design limit load is given by the solid line. The weight required to meet the creep criterion - failure after 100 hours at temperature and limit load - is given by the dashed line. Since the creep criterion requires less weight than the strength criterion throughout the temperature range of the aluminum alloy, creep is not a design problem for this material. A similar situation exists for Inconel X for most of the temperature range, but for temperatures above 1,200° F creep requires more weight than ultimate strength. Creep thus becomes important for Inconel X structures only in the temperature range where strength is decreasing rapidly with increasing temperature. For the titanium alloy, creep becomes a design problem at temperatures above 500° F. Note that above the temperature where creep becomes important, the weight required by the creep criterion increases rapidly with temperature, and design efficiency would best be obtained by conversion to a material for which creep would not be a problem. This particular titanium alloy has poor creep characteristics; others are better in creep than RC-130A, but they have less tensile strength. Further development of titanium alloys will probably improve this situation.

Since the creep criterion used in this analysis usually required less weight than the ultimate-strength criterion, it appears that creep may have little effect on aircraft structural design. In those cases where creep is a significant factor, conversion to another material will be desirable in the interest of structural efficiency and the task of designing for creep will be avoided. The results presented were obtained with a very conservative estimate of the load-temperature-time experience of the structure. If something less than 100 hours had been used for the lifetime, creep would appear to be even less important.

The importance of creep is also influenced by the type of creep criterion used, and other factors such as the permanent deformations due to creep and the effect of creep on ultimate strength should be considered, but indications are that these will be secondary design considerations. Creep failure was used as a design criterion in this analysis because of its obvious significance. A prescribed amount of permanent creep deformation could have been used, but the validity of such a criterion applied to an aircraft structure is extremely doubtful. This doubt arises from investigations of the creep behavior of aluminum-alloy structural elements and box beams coupled with the expected load-temperature-time experiences of aircraft structures. Creep may also affect the ultimate strength of the structure, but some exploratory tests on aluminum-alloy plates and box beams indicate that in many cases this effect will be negligible.

To summarize the remarks on creep, it appears that in general creep may not be a primary factor in the design of high-speed aircraft because of the character of the relationships between loads, temperatures, and time. Some materials are much more susceptible to creep than others; thus, in some cases a change in materials may be necessary in order to avoid creep problems.

THERMAL BUCKLING

Thermal buckling promises to be the most serious problem associated with the transient aspects of aerodynamic heating. Thermal buckling has an adverse effect on the bending stiffness of beams as shown in reference 6, thermal stresses less than those required for buckling reduced the effective stiffness of a cantilever plate as reported in reference 7, and aerodynamic heating and thermal stresses have incited flutter of some wing structures as described in reference 8. The work of reference 8 is now reviewed in connection with more recent tests.

In a continuation of the research described in reference 8, most models tested at Mach number 2 in the preflight jet of the Langley Pilotless Aircraft Research Station at Wallops Island, Va., experienced no difficulties, but a few came to a sudden and violent end. Failure was usually preceded by a new type of flutter, called chordwise flutter, but this flutter did not begin until the aerodynamic heating had taken effect. Chordwise flutter involves distortion of the airfoil section into a mode having about $1\frac{1}{2}$ waves along the chord - a flag-waving action. This flutter may look like panel flutter, but the complete cross section is involved rather than individual skin panels.

From the first few tests in this program, it was concluded that flutter started after the model skin buckled. However, in other tests of similar models flutter was obtained without buckling, and the concept of reduced stiffness resulting from thermal stresses has been used to explain these results. For example, a model identical to model MW-1 of reference 8 began to flutter without any evidence of buckling and fluttered for about 6 seconds before the rivets flew out and general destruction began. Model MW-4 of reference 8 and an identical model tested subsequently each fluttered and failed without any indication of buckling. This latter model had previously survived a test (at low stagnation temperature) in which it did not experience aerodynamic heating.

From the above it is evident that thermal stresses lower than those required for buckling can induce dangerous aeroelastic effects. This situation is not so serious as it seems, however, since test results show that small changes in the stiffness of the wing structure, such as the addition of a rib or two, prevented flutter without much increase in weight. (For example, a model similar to model MW-4 of ref. 8 but incorporating a single rib midway between tip and root survived the test conditions without difficulty.) Thus, careful design can prevent many failures of the type discussed above. The use of internal construction that minimizes thermal stresses may be required in some cases, but there are a wide variety of flight conditions in which more conventional designs are satisfactory.

The flight regimes in which multiweb construction may be satisfactory can be determined from the conditions that produce thermal buckling of the wing skin. Figure 3 shows some combinations of dimensions of a simplified steel multiweb beam that make the structure subject to thermal buckling (thermal stress σ_T equal to buckling stress σ_{cr}) under symmetrical aerodynamic heating conditions. The distance between webs (on a logarithmic scale) is plotted against Mach number for three altitudes (sea level, 50,000 feet, and 100,000 feet). For web spacings above a solid line, thermal buckling of the skin will occur, whereas, below the solid line, buckling will not occur.

A low level of thermal stress may be required to prevent critical changes in aeroelastic characteristics or to prevent buckling in the presence of wing bending loads. Present knowledge does not permit an estimate of the aeroelastic effects of a given thermal stress distribution or of the level of thermal stress that may be tolerated, but an assumption that thermal stresses only 10 percent of those required for buckling may be permissible appears to be reasonable. The dash-line curves give the web spacings corresponding to this 10-percent condition.

The web spacings that would be used for an efficient load-carrying structure would be around 2 to 5 inches; thus such a structure would be in no danger of buckling due to heating alone, but a low level of thermal stress could not be attained in this structure at Mach numbers above 3 without very closely spaced webs.

Figure 3 is based on several approximations (see derivation of equations in appendix B) and a simplified structure and thus indicates only the approximate combinations of dimensions and flight conditions that will produce buckling of one series of beams. Similar charts are easily prepared for other configurations, but an actual design should be checked by a more exact analysis.

In summarizing the remarks on thermal buckling, the present state of knowledge indicates that thermal buckling should be prevented. In many cases thermal stresses well below those required for buckling may lead to dangerous aeroelastic effects. The proper location of internal members may prevent such effects without undue weight penalties under many flight conditions, but special types of internal structure that minimize thermal stresses will be necessary when severe aerodynamic heating is encountered.

INSULATION

Having discussed two of the problems resulting from aerodynamic heating, consideration is now given to one of the ways of alleviating heating effects, namely, insulating the structure. One of the many probable uses of insulation is to extend the aerodynamic-heating conditions under which aluminum alloys are useful structural materials. Figure 4 presents results to show the effect of insulation on the temperature history of a 0.10-inch-thick skin of aluminum alloy.

Temperature T is plotted against time in minutes for an instantaneous acceleration to a Mach number of 4 at an altitude H of 50,000 feet. The temperature history of the uninsulated skin is shown by the upper solid line and the effect of 0.1 inch of insulation such as rock wool or asbestos (thermal conductivity of 0.03 Btu/ft-hr-$^{\circ}$F) is shown by the lower solid line. The temperature of the outer surface of the insulation is given by the dashed line. The equations used to calculate these curves, and others to be presented illustrating the effect of insulation, are given in appendix C.

For the flight condition used in figure 4, the aluminum alone would quickly reach a temperature at which its strength had vanished, whereas a relatively thin layer of insulation holds the temperature down for several minutes. This is a very beneficial result insofar as the structure is concerned, but this benefit has been obtained by transferring difficult design problems to the insulation. The temperature of the heated surface of the insulation rises very quickly to a value near the equilibrium temperature T_{eq} (which is assumed equal to the adiabatic wall temperature T_{AW}) and thereafter changes very slowly. The insulation must thus withstand high temperatures and large temperature differences which will require special design techniques for the construction and attachment of such a protective coating.

The fact that the surface temperature of the insulation closely approximates the equilibrium temperature permits considerable simplification of the relation between the insulation required and the flight condition. By neglecting the specific heat of the insulation and assuming that the temperature of the outer surface of the insulation is the equilibrium temperature, a conservative estimate of the required thickness of insulation can be obtained, as shown in figure 5 (see eq. (C7)). The required thickness is plotted against surface temperature for 0.10-inch-thick skin of aluminum that rises from 0° to 200° F in the times indicated. Each of the curves is for a particular value of the time required for the structure to reach 200° F and the thermal conductivity of the insulation is assumed to be invariant with temperature. This figure shows that high equilibrium temperatures and long flight times require very thick layers of insulation, thicknesses that are impractical for many parts of the aircraft such as thin wings or control surfaces.

The situation improves somewhat for heavier structures since the same curves apply to other thicknesses of aluminum skin. For example, if the thickness of the skin is doubled, the time during which the insulation is effective will also be doubled or the required insulation thickness will be cut in half. The required thickness is substantially reduced if the skin is allowed to go to higher temperatures, but for this case the relationship is not so simple. Higher skin temperatures would necessitate another structural material and lead to considerations of structural efficiency.

The thicknesses indicated in figure 5 are rather large and somewhat discouraging; however, another approach to the insulation problem that can result in lower heat transfer, less thickness, and possibly lighter weight is the use of radiation shields.

Average heat transfer through two types of insulators is shown in figure 6 as a function of the surface equilibrium temperature. The solid lines are for two thicknesses l of the previously discussed bulk insulation such as rock wool or asbestos. The dashed lines are for a single radiation shield at two values of emissivity ϵ. The radiation shield would consist of a thin sheet of metal supported a short distance away from the outer surface of the structure. The aerodynamic surface of the shield would be a black-body radiator (high emissivity) to keep surface temperature low, but the inner surface of the shield and outer surface of the structure would be very bright (low emissivity) to minimize radiant heat exchange. Some low-conductivity supporting structure, such as a honeycomb core, would be needed between them, but in this analysis, the heat transferred by this supporting structure and the air between shield and structure has been neglected (see appendix C). The emissivity of 0.1 is about the upper limit for most shiny metals, but the value of 0.02 can be attained only with special coatings such as gold or silver plate.

The average heat flow plotted is the average rate at which the insulators transfer heat to the structure while the structure rises from 0° to 200° F. The conditions considered are similar to those in figure 5, but by plotting average heat flow, all combinations of time and heat capacity of the structural material are included. Where two curves cross, the insulators are exactly equivalent for the conditions considered. At other points, the lowest curve indicates the best insulator. The curves show that the radiation shield provides excellent protection at the lower temperatures but becomes increasingly inefficient as the temperature increases. The heat flow through the shield can be further reduced by using several of them; for example, five shields of 0.1 emissivity would transfer about the same amount of heat as the single shield of 0.02 emissivity, but the added complication of multiple shield arrangements might prove to be impractical.

An effective radiation shield requires bright surfaces that are attainable only with metals at moderate temperatures; thus this approach would probably be limited to temperatures below about 1,500° F. Above this temperature, a type of bulk insulation suitable for high-temperature use would be required. For some applications, such as protection of occupants, a combination of insulation, radiation shields, and cooling may provide the most efficient combination.

The results presented indicate that insulation provides a suitable way to alleviate the effects of aerodynamic heating on the primary structure under certain flight conditions. This alleviation can be attained, however, only by creation of many difficult problems associated with the design, fabrication, and installation of the protective coatings.

CONCLUDING REMARKS

The problems of creep and thermal buckling have been discussed in an effort to indicate their effects on the design of high-speed aircraft. The results have shown that creep may not be a significant factor, but thermal buckling may have a substantial effect on the structural design. Consideration has also been given to the use of insulation for the alleviation of heating effects; such an approach has merit under certain conditions but poses many new design problems.

The results presented are based on the limited existing information on the structural problems of aerodynamic heating. The picture may change as more research results become available, but the design problems undoubtedly will be eased as knowledge increases. It was not intended to convey the impression that aerodynamic-heating problems are easily overcome but merely to indicate how careful design can solve some of them.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., June 2, 1955.

APPENDIX A

SYMBOLS

A	cross-sectional area, sq ft
b	web spacing, ft
c	specific heat, Btu/lb-°R
E	modulus of elasticity, lb/sq ft
H	altitude, ft
h	heat-transfer coefficient, Btu/ft^2-sec-°R
k	thermal conductivity, Btu/ft-sec-°R
L	web depth, ft
l	insulation thickness, ft
M	Mach number
N	number of radiation shields
n	load factor
n_L	limit load factor
q	rate of heat transfer, Btu/ft^2-sec
T	temperature, °R
t	thickness, ft
w	specific weight, lb/cu ft
z	distance along web from center line, ft
α	coefficient of expansion, ft/ft-°R

$$\gamma = \sqrt{\frac{hL^2}{4kt_s}}$$

ϵ emissivity

μ Poisson's ratio

σ Stefan-Boltzmann constant, Btu/ft^2-sec-$^{\circ}$R^4

σ_T thermal stress, lb/sq ft

σ_{cr} buckling stress, lb/sq ft

τ time, sec

Subscripts:

AW adiabatic wall

eq equilibrium

i insulation

o initial condition

s skin

w web

A bar over a symbol indicates an average value.

The units given above are those required to keep the equations in appendixes B and C dimensionally correct and do not necessarily apply to the quantities used in the figures.

APPENDIX B

THERMAL BUCKLING OF AERODYNAMICALLY HEATED MULTIWEB BEAMS

For the simplified type of multiweb construction shown in figure 3, the maximum compressive stresses (thermal) occur in the skin when the wing is subjected to symmetrical aerodynamic heating. The thermal stresses in the skin are given by

$$\sigma_T = E\alpha(\bar{T} - T_s) \tag{B1}$$

where $\bar{T}$ is the average temperature of the cross section; that is,

$$\bar{T} = \frac{A_s\bar{T}_s + A_w\bar{T}_w}{A_s + A_w} \tag{B2}$$

and

$$A_s = 2bt_s$$

$$A_w = Lt_w$$

The temperature distribution resulting from instantaneous acceleration to a given flight condition can be obtained from the analysis of reference 11 or 12, but the equations are complex and somewhat difficult to apply. In appendix I of reference 2, however, equations for the temperature distribution are presented that are a good approximation and are easy to use since they involve only one parameter in addition to the usual temperature and time parameters. In this approximate analysis, it is assumed that the skin temperature is uniform at each instant of time and given by

$$T_s = T_{AW} - (T_{AW} - T_o)e^{-\frac{h\tau}{cwt_s}} \tag{B3}$$

The web temperature at $z = \frac{L}{2}$ follows the above relation, and the temperature distribution in the web is found by solving the equation of heat conduction. The result from appendix I of reference 2 (with the notation changed to that of this paper) is

$$T_W = T_O + (T_{AW} - T_O)\left[1 - \frac{\cos\frac{2\gamma z}{L}}{\cos\gamma} e^{-\frac{h\tau}{cwt_s}} - \frac{4}{\pi}\sum_{n=1,3,5}^{\infty} \frac{(-1)^{\frac{n-1}{2}}}{n}\left(\frac{\cos\frac{n\pi z}{L}}{1 - \frac{n^2\pi^2}{4\gamma^2}}\right) e^{-\frac{n^2\pi^2}{4\gamma^2}\frac{h\tau}{cwt_s}}\right] \tag{B4}$$

where

$$\gamma = \sqrt{\frac{hL^2}{4kt_s}}$$

The above solution converges if $\gamma^2 \neq \frac{n^2\pi^2}{4}$. In reference 11, it is shown that the thermal stresses obtained from this temperature distribution agree well with the exact analysis and that the results for an instantaneous acceleration are only slightly conservative when compared with results for a finite acceleration.

With the above assumptions the uniform thermal stress in the skin is given by

$$\sigma_T = \frac{-E\alpha(T_s - \overline{T}_w)}{1 + \frac{A_s}{A_w}} \tag{B5}$$

and

$$\frac{T_s - \overline{T}_w}{T_{AW} - T_o} = \left(\frac{\tan \gamma}{\gamma} - 1\right)e^{-\frac{h\tau}{cwt_s}} + \frac{8}{\pi^2}\sum_{n=1,3,5}^{\infty}\frac{e^{-\frac{n^2\pi^2}{4\gamma^2}\frac{h\tau}{cwt_s}}}{n^2\left(1 - \frac{n^2\pi^2}{4\gamma^2}\right)} \tag{B6}$$

The thermal stresses vary with time, but those in the skin have a maximum at $\tau = \tau'$ when

$$\left(1 - \frac{\tan \gamma}{\gamma}\right)e^{-\frac{h\tau'}{cwt_s}} = 2\sum_{n=1,3,5}^{\infty}\frac{e^{-\frac{n^2\pi^2}{4\gamma^2}\frac{h\tau'}{cwt_s}}}{\gamma - \frac{n^2\pi^2}{4}} \tag{B7}$$

The maximum temperature difference of equation (B6) and the time at which it occurs have been evaluated as functions of γ^2 and are plotted in figure 7.

The thermal stresses given by equation (B5) can now be made any desired fraction of the skin buckling stress. For purposes of this analysis, it is assumed that the following elastic-buckling formula for a simply supported plate applies:

$$\sigma_{cr} = -\frac{4\pi^2 E}{12(1 - \mu^2)}\left(\frac{t_s}{b}\right)^2 \tag{B8}$$

The above equation should give satisfactory results in the elastic range, but if thermal buckling occurs at stress levels in the plastic range neither equation (B5) nor (B8) is strictly applicable. There is little known about thermal buckling under such conditions; consequently, for the approximate analysis of this report it will be assumed that the elastic analysis applies throughout the stress range since the errors in equations (B5) and (B8) may compensate one another.

To obtain the curves in figure 3, heat-transfer coefficients were determined from reference 13 and the following properties of the steel multiweb beams were used:

$$L = 4 \text{ in.}$$

$$t_s = 0.125 \text{ in.}$$

$$t_w = 0.031 \text{ in.}$$

$$\mu = 0.33$$

$$\alpha = 7.6 \times 10^{-6} \text{ in./in.-}^{\circ}\text{R}$$

$$c = 0.11 \text{ Btu/lb-}^{\circ}\text{R}$$

$$w = 500 \text{ lb/cu ft}$$

$$k = 8.5 \text{ Btu/ft-hr-}^{\circ}\text{R}$$

APPENDIX C

EFFECTS OF INSULATION

Temperature Histories

If an instantaneous acceleration to a given flight condition is assumed, the temperature of a thin uninsulated metal skin is given by

$$T_s = T_{AW} - (T_{AW} - T_o)e^{-\frac{h\tau}{cwt_s}} \tag{C1}$$

If the heat capacity of the insulation is neglected, equation (C1) applies to the insulated metal skin if the heat-transfer coefficient is replaced by a value that includes the thermal resistance of the insulation. Thus

$$\frac{1}{h'} = \frac{1}{h} + \frac{l}{k} \tag{C2}$$

$$T_s' = T_{AW} - (T_{AW} - T_o)e^{-\frac{h'\tau}{cwt_s}} \tag{C3}$$

Corresponding to the above results, the temperature of the exposed surface of the insulation is given by

$$T_i = \frac{T_s' + \frac{hl}{k} T_{AW}}{1 + \frac{hl}{k}} \tag{C4}$$

where T_s' varies with time as given by equation (C3). Equation (C4) indicates that when the skin is well insulated (k/l is small compared with h) T_i is very nearly equal to T_{AW}, a fact that is used in the next section. Also, according to equation (C4), T_i starts at a temperature above the initial temperature of the system because the heat capacity of insulation was neglected, but an approximate indication of initial variation of the surface temperature of the insulation can be obtained from

$$T_i' = T_i - (T_i - T_o)e^{-\frac{2h\tau}{c_i w_i l}} \tag{C5}$$

where T_i is given by equation (C4).

The curves in figure 4 were calculated by using the above equation and the following numerical values:

For aerodynamic heating,

$$T_{eq} = T_{AW} = 1{,}016^{\circ}\ \mathrm{F}$$

$$T_o = -67^{\circ}\ \mathrm{F}$$

$$h = 0.011\ \mathrm{Btu/ft^2\text{-}sec\text{-}^{\circ}R}$$

for the aluminum skin,

$$c = 0.23\ \mathrm{Btu/lb\text{-}^{\circ}R}$$

$$w = 173\ \mathrm{lb/cu\ ft}$$

$$t_s = 0.10\ \mathrm{in.}$$

and for the insulation,

$$c_i = 0.20\ \mathrm{Btu/lb\text{-}^{\circ}R}$$

$$w_i = 20\ \mathrm{lb/cu\ ft}$$

$$l = 0.01\ \mathrm{in.}$$

$$k = 0.030\ \mathrm{Btu/ft\text{-}hr\text{-}^{\circ}R}$$

Insulation Required

A simple equation for the required thickness of insulation can be obtained by neglecting the heat capacity of the insulation and assuming that the temperature of the exposed surface of the insulation is the equilibrium temperature. Then the heat-balance equation

$$\frac{k}{l}(T_{eq} - T_s) = cwt_s \frac{dT_s}{d\tau} \tag{C6}$$

has the following solution:

$$T_s = T_{eq} - (T_{eq} - T_o)e^{-\frac{k\tau}{cwt_s l}} \tag{C7}$$

The required thickness is given by

$$l = \frac{k\tau}{cwt_s} \Bigg/ \log_e\left(\frac{T_{eq} - T_o}{T_{eq} - T_s}\right) \tag{C8}$$

Equation (C8) is accurate if the heat capacity of the insulation is small compared with the heat capacity of the metal skin, but it is very conservative if the heat capacities are of the same order of magnitude.

The curves in figure 5 were calculated by using equation (C8) and the following numerical values:

$$T_o = 0^\circ \text{ F}$$

$$T_s = 200^\circ \text{ F}$$

$$k = 0.030 \text{ Btu/ft-hr-}^\circ\text{R}$$

$$c = 0.23 \text{ Btu/lb-}^\circ\text{R}$$

$$w = 173 \text{ lb/cu ft}$$

$$t_s = 0.10 \text{ in.}$$

Radiation Shield

The heat transferred by radiation between two parallel plane surfaces of infinite extent is given by (ref. 14)

$$q = \frac{\sigma}{\frac{1}{\epsilon_1} + \frac{1}{\epsilon_2} - 1}\left(T_1^{\,4} - T_2^{\,4}\right) \tag{C9}$$

If the metallic skin is protected by N radiation shields that are gray-body radiators of emissivity ϵ and the inside surface of the metallic skin has this same emissivity, then the heat transferred to the structure by radiation is

$$q = \frac{\sigma}{N\left(\frac{2}{\epsilon} - 1\right)}\left(T_{eq}^{\,4} - T_s^{\,4}\right) \tag{C10}$$

The time required for the structure to reach a prescribed temperature is given by the following equation which was obtained by neglecting the heat capacity of the radiation shields and assuming that the structure received heat only by radiation:

$$\tau = \frac{N(2 - \epsilon)}{4\sigma\epsilon T_{eq}^{\,3}}\,\Omega c w t_s \tag{C11}$$

where

$$\Omega = \log_e \frac{T_{eq} + T_s}{T_{eq} - T_s} + 2\tan^{-1}\frac{T_s}{T_{eq}} - \log_e \frac{T_{eq} + T_o}{T_{eq} - T_o} - 2\tan^{-1}\frac{T_o}{T_{eq}}$$

Average Heat Flow

A convenient way to compare the bulk type of insulation with the radiation shield is to consider the average rate of heat transfer during the time τ that the skin is rising from the initial temperature T_o to the final temperature T_s. This average rate of heat transfer $\bar{q}$ is then defined as follows:

$$\bar{q} = \frac{cwt_s}{\tau}(T_s - T_o) \tag{C12}$$

For the bulk insulation, by using equations (C8) and (C12), the result is

$$\bar{q} = \frac{\frac{k}{l}(T_s - T_o)}{\log_e\left(\frac{T_{eq} - T_o}{T_{eq} - T_s}\right)} \tag{C13}$$

and for the radiation shields, by using equations (C11) and (C12), the result is

$$\bar{q} = \frac{4\sigma T_{eq}^{3}(T_s - T_o)}{\Omega N\left(\frac{2}{\epsilon} - 1\right)} \tag{C14}$$

The curves of figure 6 were calculated by using equations (C13) and (C14) and the following numerical values:

$T_o = 0^\circ$ F

$T_s = 200^\circ$ F

$k = 0.030$ Btu/ft-hr-°R

$\sigma = 0.173 \times 10^{-8}$ Btu/ft²-hr-°R

$N = 1$

REFERENCES

1. Anon.: Symposium on the Thermal Barrier. Presented by the Aviation Division of the ASME at the annual meeting, Nov. 28 - Dec. 3, 1954.

2. Hoff, N. J.: Structural Problems of Future Aircraft. Third Anglo-American Aeronautical Conference, Brighton, Joan Bradbrooke and E. C. Pike, eds., R.A.S., 1952, pp. 77-114.

3. Hoff, N. J.: The Structural Effects of Aerodynamic Heating. PIBAL Rep. No. 240 (Contract No. AF 33(616)-116), Jan. 1954.

4. Loveless, E., and Boswell, A. C.: The Problem of Thermal Stresses in Aircraft Structures. Aircraft Engineering, vol. XXVI, no. 302, Apr. 1954, pp. 122-124.

5. Hibbard, Hall L., and McBrearty, J. F.: Structures for High Speed Aircraft. Aviation Age, vol. 20, no. 5, Nov. 1953, pp. 32-51; Structures for High-Speed Flight - II. Aviation Age, vol. 20, no. 6, Dec. 1953, pp. 116-123.

6. Pride, Richard A.: An Investigation of the Effects of Rapid Skin Heating on Box Beams Loaded in Bending. NACA RM L55B03, 1955.

7. Vosteen, Louis F., and Fuller, Kenneth E.: Behavior of a Cantilever Plate Under Rapid Heating Conditions. NACA RM L55E20c, 1955.

8. Heldenfels, Richard R., and Rosecrans, Richard: Preliminary Results of Supersonic-Jet Tests of Simplified Wing Structures. NACA RM L53E26a, 1953.

9. Press, Harry, Meadows, May T., and Hadlock, Ivan: Estimates of Probability Distribution of Root-Mean-Square Gust Velocity of Atmospheric Turbulence From Operational Gust-Load Data by Random-Process Theory. NACA TN 3362, 1955.

10. Mayer, John P., and Hamer, Harold A.: A Study of Means for Rationalizing Airplane Design Loads. NACA RM L55E13a, 1955.

11. Parkes, E. W.: Transient Thermal Stresses in Wings. Aircraft Engineering, vol. XXV, no. 298, Dec. 1953, pp. 373-378.

12. Pohle, Frederick V., and Oliver, Henry: Temperature Distribution and Thermal Stresses in a Model of a Supersonic Wing. Jour. Aero. Sci., vol. 21, no. 1, Jan. 1954, pp. 8-16.

13. Van Driest, E. R.: Turbulent Boundary Layer in Compressible Fluids. Jour. Aero. Sci., vol. 18, no. 3, Mar. 1951, pp. 145-160, 216.

14. Jakob, Max, and Hawkins, George A.: Elements of Heat Transfer and Insulation. Second ed., John Wiley & Sons, Inc., c.1942, 1950, p. 184.

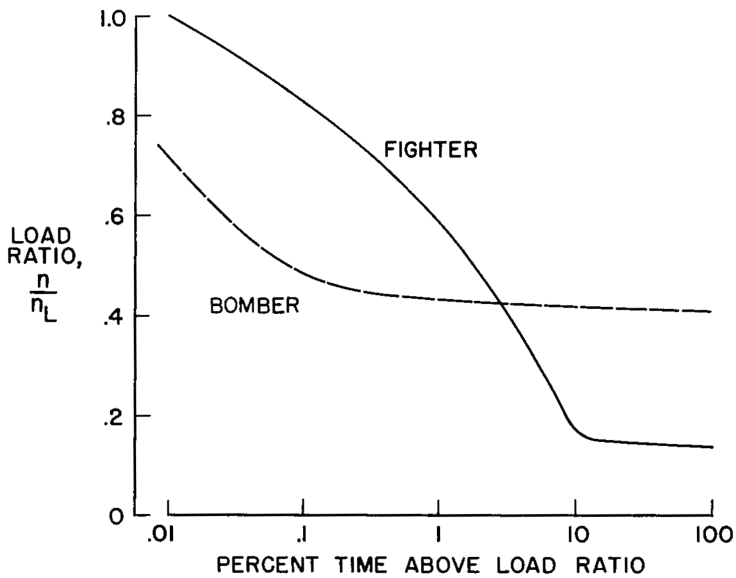

Figure 1.- Load-time relations for fighter- and bomber-type airplanes.

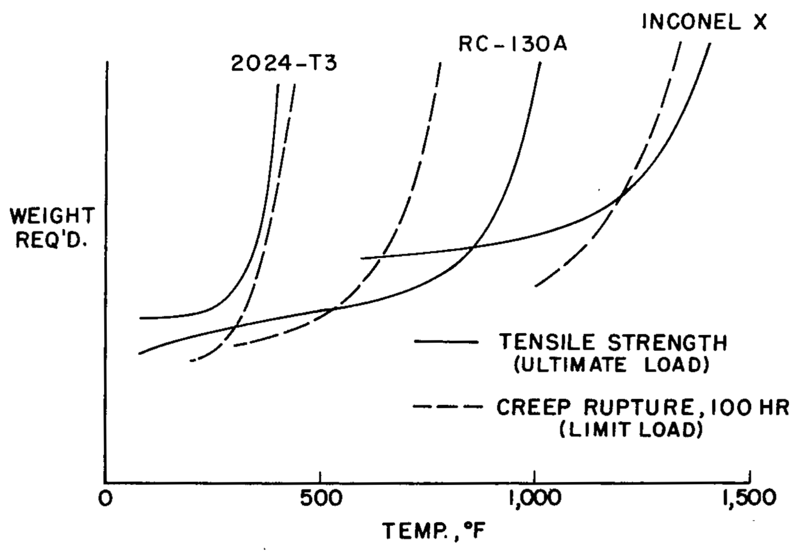

Figure 2.- Comparison of weight required to support a constant tensile load for two design criteria.

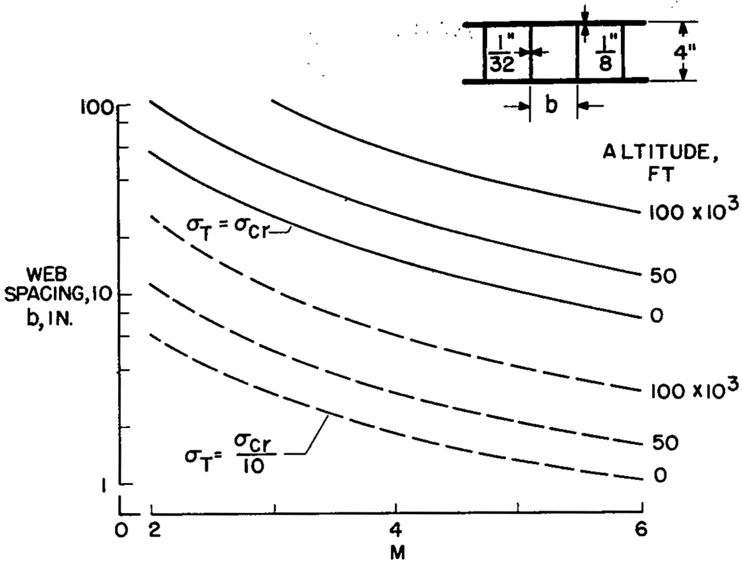

Figure 3.- Thermal-buckling boundaries for a steel multiweb beam.

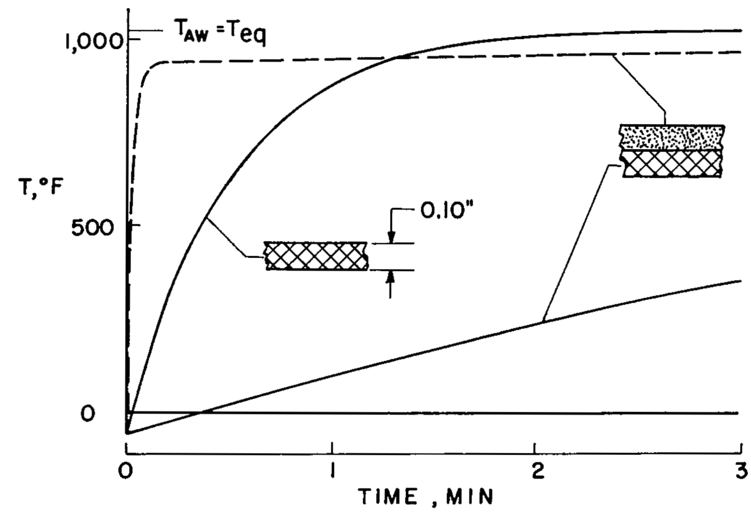

Figure 4.- Temperature histories of insulated and uninsulated aluminum at $M = 4$ and an altitude of 50,000 feet.

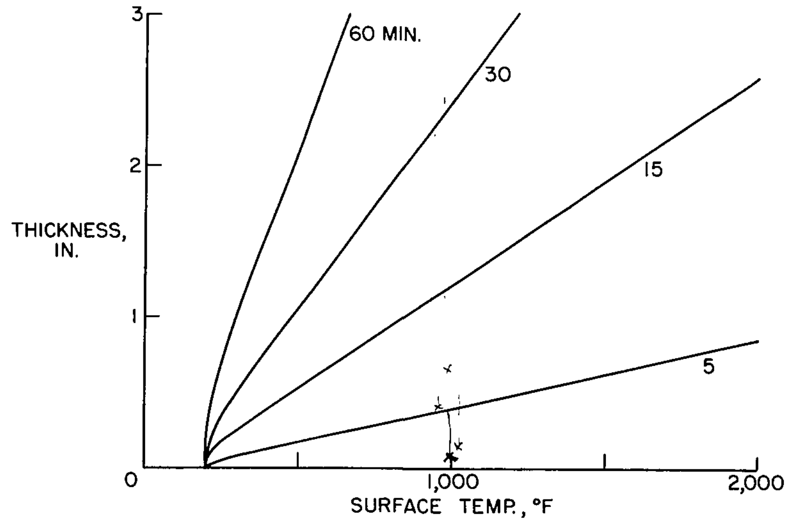

Figure 5.- Insulation required to limit the temperature of 0.10-inch aluminum to a rise from 0° to 200° F.

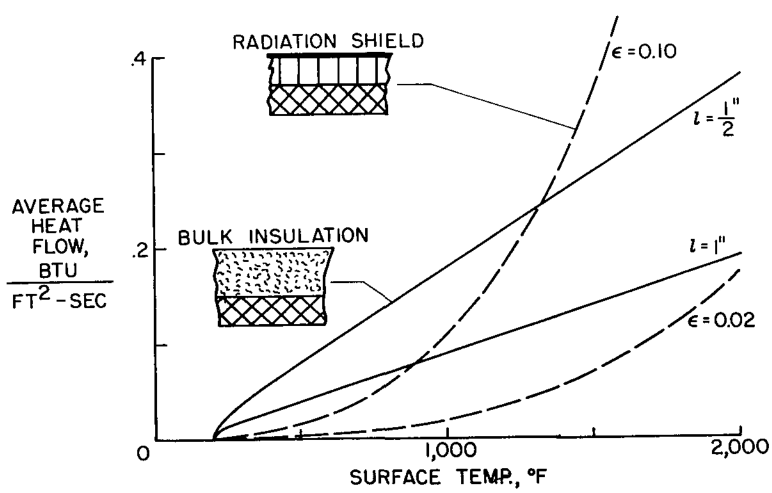

Figure 6.- Comparison of the average heat flow through two types of insulators.

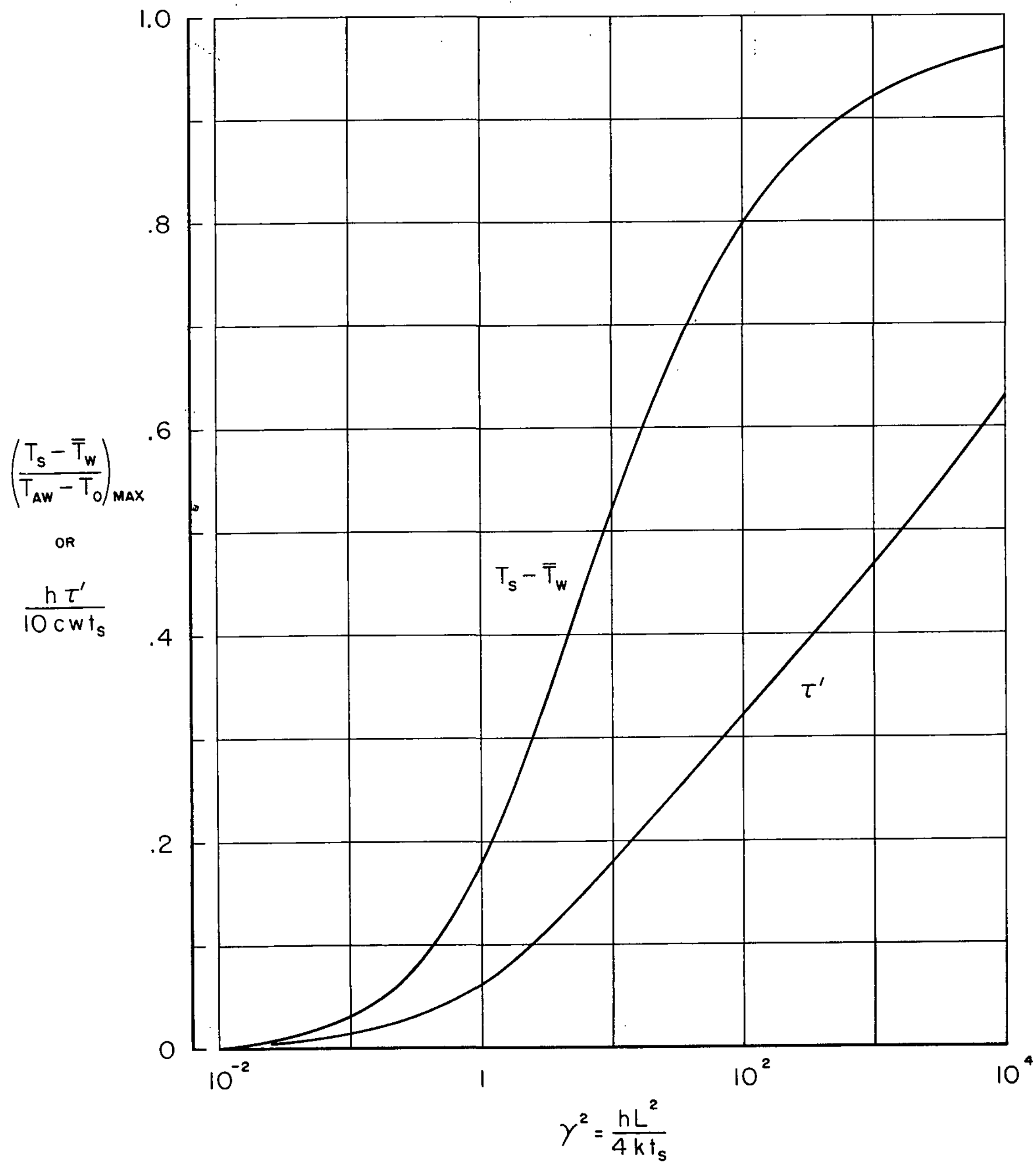

Figure 7.- Chart for determining the time and magnitude of the temperature difference that produces maximum thermal stresses in the skin of an aerodynamically heated multiweb beam.